

SIMONE: Interplanetary Microsatellites for NEO Rendezvous Missions

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Abstract

The paper summarises a novel mission concept called SIMONE (Smallsat Intercept Missions to Objects Near Earth), whereby a fleet of microsatellites may be deployed to individually rendezvous with a number of Near Earth Objects (NEOs), at very low cost. The mission enables, for the first time, the diverse properties of a range of spectral and physical type NEOs to be determined. Such data are invaluable in the NEO scientific, impact damage prediction, and impact countermeasure planning contexts. The five identical 120kg spacecraft are designed for low-cost piggyback launch on the Ariane-5 into GTO, from where each uses a high specific impulse gridded-ion engine to escape Earth gravity and ultimately to rendezvous with a different NEO target. The primary challenge with such a mission is the ability to accommodate the necessary electric propulsion, power, payload and other onboard systems within the severe constraints of a microsatellite. The paper describes the way in which the latest technological advancements and innovations have been selected and applied to the mission design. The SIMONE design is feasible and offers a highly cost-effective mission, which is applicable to other solar system science/exploration objectives. SIMONE clearly demonstrates that the concept of an “interplanetary microsatellite” is now realisable.

Introduction

Exploration of the solar system, including small bodies such as asteroids, is a very demanding application of space technology. It has traditionally required a dedicated high energy launch and also a spacecraft with a significant propulsive capability. Such missions can therefore be both large and costly, although medium-sized missions have also flown at a moderate cost recently¹. However, truly low-cost access to the frontier is now within reach with the advent of the “interplanetary microsatellite”.

Recent developments in key enabling technologies have been brought together to form a high-performance microsatellite bus design that has a significant delta-V capability of 10km/s, whilst carrying a number of miniaturised but capable payload instruments. The key enabling technology is low-thrust, very high-specific impulse ion propulsion. Recently validated by the Deep Space 1 mission², ion propulsion combines a high delta-V capability with very low fuel consumption. By integrating this

emerging technology into a microsatellite system of <120kg mass, solar system exploration such as asteroid rendezvous, can be done for a fraction of the cost of a traditionally larger, chemically propelled mission. The microsatellite must generate high power to drive the ion engine. This power can be made available to the payload and on-board communications system for the data downlink to a commercial deep space network provider, when the ion engine is not operating.

The small-sized bus substantially reduces mission cost by enabling cheap access to space via piggyback or ride-sharing launch opportunities into high Earth orbit, and a rapid development and build schedule. Significant design challenges arise when attempting to integrate the ion propulsion system into the confines of a small bus, and supporting its high power requirements (500-1000W). However, these difficulties have been overcome during detailed studies of the concept and feasibility of the SIMONE mission for the European Space Agency³, as described below in this paper.

The Mission

Mission Concept

SIMONE is a unique interplanetary mission concept comprising a fleet of low-cost microsatellites that will individually rendezvous with a different Near Earth Object (NEO), each of a distinct spectral and/or physical type. In-situ science measurements taken by instruments on-board each spacecraft enable the wide diversity in the physical and compositional properties of the NEO population to be characterised in a highly cost-effective manner. Analysis of the in-situ measurement data will provide:

- valuable scientific knowledge on the nature, origin and processing of NEOs;
- critical physical/compositional information needed for the accurate prediction of impact risk (particularly damage potential) posed by NEOs;
- critical physical/compositional information needed for the development of effective NEO risk mitigation strategies that are specifically tailored for each NEO type.

The SIMONE mission team is led by QinetiQ – formerly DERA - (UK) in partnership with the Planetary and Space Sciences Research Institute (PSSRI) of the Open University (UK), with additional expertise provided by SciSys (UK), Politecnico di Milano (Italy) and Telespazio (Italy).



Figure 1. The SIMONE microsatellite during rendezvous with a NEO target (image of Ida courtesy of NASA)

Mission Overview

SIMONE can be realised by the use of microsatellite technology – this would be a world-first for an interplanetary mission. Conducting multiple NEO rendezvous missions with large conventional spacecraft would be prohibitively expensive. It is

planned to deploy 5 of the SIMONE microsatellites to their intended rendezvous targets within the budget envelope of \$150M. In order to achieve this target, a low-cost approach to the programme as a whole is needed.

The SIMONE microsatellites (see Figure 1) are based around a single spacecraft system design, configuration and payload, and a single ground segment, thereby significantly lowering recurring costs. “Piggyback” launch opportunities on the Ariane Structure for Auxiliary Payloads (ASAP) on Ariane-5 will be exploited in order to obtain low launch costs (see Figure 2). Traditionally, launch costs are a significant cost driver for interplanetary missions because a dedicated deep space launch is usually required for direct injection onto an interplanetary trajectory. Instead, these launches will place each SIMONE spacecraft into a Geostationary Transfer Orbit (GTO). From GTO, on-board propulsion is used to achieve an Earth escape trajectory, adjust the interplanetary trajectory, eventually rendezvous with the target NEO and conduct close reconnaissance around the target.



Figure 2. QinetiQ STRV microsatellites mounted on the Ariane-5 ASAP destined for GTO

Mission Constraints

The main design constraints placed on such a low-cost mission arise from the build, launch and operations elements of the project. The development schedule, from detailed design to launch, is foreseen to be no more than 4 years for all 5 spacecraft. For launch as a piggyback payload on an Ariane 5, the mass of each spacecraft is limited to 120kg. The available volume for a single spacecraft bus on the launcher is a box of no more than 600x600x710mm, with a negotiated waiver for the stowed solar array wings of 200x300x700mm either side of the bus. As an auxiliary passenger, there is also no influence over the launch date/window, requiring a flexible mission design strategy. The total 5-flight mission duration is limited to 5 years in order to reduce ground segment costs and space segment component costs.

Table 1: Summary of the SIMONE mission phases

	Mission Phase	Duration	Description
1	Launch and Early Orbit Phase (LEOP)	~3-4 days	Deployment from an Ariane-5 launch vehicle into GTO, and attainment of a stable 3-axis attitude control mode following ejection from the Ariane upper stage
2	Check Out Phase	~2 weeks	Complete system functional tests during visibility periods with the ground station.
3	Parking Orbit Phase	~9-12 months (depending upon time to optimum Escape Phase start)	From highly eccentric GTO, raising of the orbit above the proton & electron radiation belts into a safe (i.e. low radiation dose) near-circular “parking orbit” at 300,000km. Wait for the Earth-Escape Phase to start.
4	Earth Escape Phase	~1 month	Expand and phase the orbit using the on-board ion propulsion system for a lunar swing-by gravity assist manoeuvre to lower delta-V and propellant consumption. Exceed Earth gravitational sphere of influence into a heliocentric orbit with an inclination similar to the target plane.
5	Rendezvous Phase	~22-33 months (depending upon the NEO target orbit)	Combination of phased long-duration low thrust arcs and coast (no thrust) arcs in heliocentric orbit. Follow an optimised transfer trajectory that ultimately arrives in vicinity of the target at a low approach velocity and matches the spacecraft’s orbit with that of the target NEO orbit. Trajectory correction manoeuvres are determined based upon radio navigation techniques. Acquisition of the target NEO using the imager payload for optical navigation relative to the target. Long-range approach to the target until reaching a stand-off distance from 2000km down to ~200km ready for the Measurement Phase.
6	Measurement Phase	~4 months	Intermediate proximity: <ul style="list-style-type: none"> Co-fly with the target NEO at ~100-200km range and take in-situ measurements with the payload instruments to determine size and shape. Close proximity: <ul style="list-style-type: none"> Close ballistic swing-bys of the spacecraft at a minimum distance of a few NEO radii (<10km) to determine gravity field and mass. Determine bulk density from shape, mass. “Imaging” swing-bys at <10km altitude to obtain high-resolution data on NEO surface features/composition.

Mission Design

The six different phases of each SIMONE flight are described in Table 1. Trajectory analyses were performed for each of the selected rendezvous targets by Politecnico di Milano⁴. Low-thrust trajectory optimisation software was used to determine minimum fuel mass trajectories for SIMONE arrival at the targets within the 5-year total mission duration constraint and with a low arrival velocity. These optimal trajectories require lunar swing-by and Earth escape to occur on specific dates to enable minimum fuel expenditure. However, there is no control over the launch date as the auxilliary launch passenger. Therefore, a flexible mission design strategy was devised to overcome this problem. This involves booking a launch that deploys well in advance of the required swingby/escape dates and raising the orbit from a GTO to a high altitude parking orbit. This is

done as quickly as possible to clear the radiation belts and hence minimise solar array power degradation. The parking orbit is located at 300,000km semi-major axis with eccentricity 0.25, and is obtained by firing the engine continuously with the thrust vector aligned with the velocity vector all around the orbit arc from GTO start until parking orbit stop condition. The exception is during short eclipses when the engine power requirements cannot be met by the power system. Once in the parking orbit, the spacecraft can wait until the time for optimal lunar swingby and Earth escape to begin. Being in an orbit fairly close to the Moon, it is then easy to fire the engine to phase the orbit for targeting the lunar swingby at the correct point of the hyperbolic periapsis. This gravity assist has been determined to achieve escape with a heliocentric plane change, thus reducing the demand on the engine to match the inclination of the NEO target during the interplanetary phase.

The Science

Rationale

The NEO population contains a wide diversity of bodies with different physical properties relating to their origin and subsequent processing by solar radiation, cosmic ray and impact effects. Ground-based optical telescope observations have enabled a number of different types of NEO to be characterised by their unique spectral signatures. From the observations, physical properties such as size, shape, rotation, surface mineralogical type can be inferred. Radar observations can give excellent information on orbit, rotation state, surface topography, and binary nature, with the latter being useful to accurately derive object mass.

Whilst remote observations permit classification of general compositional types and determination of some physical properties, the range of uncertainty of this inferred physical information is often large and insufficient for impact risk assessment and mitigation purposes. Furthermore, some information simply cannot be collected from ground observations. Only dedicated rendezvous missions can provide the in-depth study of NEOs that is required to fully characterise their physical and compositional properties with high accuracy. A multiple rendezvous strategy to sample the different *types* of NEO at close range addresses critical gaps in our knowledge of the NEO population that would otherwise remain, even in the light of the various forthcoming missions. Filling these knowledge gaps is fundamental to how well we can define and counter the threat posed by NEOs to the Earth in the future.

The diversity of NEO physical properties (particularly bulk density) leads to great uncertainty in the magnitude of destruction that an object of a particular size would cause on Earth. The data provided by multiple rendezvous missions would allow better predictive tools to be established that would link key properties such as bulk density with parameters that can be derived from Earth-based or space-based remote observations. Such links are important in that they would allow a better initial characterisation of a potentially threatening object (in terms of size, mass and composition), thus influencing the decision regarding an effective mitigation approach.

Mitigation techniques may involve a slight deflection of a minor body by either a carefully planned impact (or series of impacts), long-term ablation of the surface, an explosive impulse, or long-term action of a low-thrust propulsive device. Understanding key properties of the surface and bulk materials of different types of NEOs and their response to heating or mechanical stress is thus crucial to the selection and development of effective mitigation techniques.

Some techniques may be wholly ineffective for certain types of target object, such as highly porous or fractured bodies. Close-up reconnaissance of diverse NEOs is a logical precursor to the success of missions dedicated to executing specific mitigation strategies.

The multiple rendezvous philosophy echoes one of the recommendations of the report of the UK Government's Taskforce on Potentially Hazardous Near Earth Objects⁵, namely that the case for mounting a number of co-ordinated rendezvous missions to visit different types of NEO should be explored. The recommendation goes on to suggest that they be based on relatively inexpensive microsatellites. This was the starting point for the SIMONE concept.

Rendezvous Targets

An extensive target selection process has been conducted in order to identify the most suitable Near Earth Objects (NEOs) for rendezvous by five SIMONE deep space missions, according to priorities for NEO risk assessment, science and mission feasibility. Filtering techniques were applied to the known NEO population of nearly 1,927 objects as of 2nd July 2002. These filters were designed to reduce the population down to a shortlist of potential candidates, where the feasibility of a SIMONE rendezvous mission could be ensured in terms of accessibility, navigation to the targets, and achieving science objectives at the objects. The primary filter criteria applied to the population were:

- **Delta-V** to rendezvous with the NEO targets, after the Earth escape phase, must be within the remaining delta-V capability of the SIMONE spacecraft (~6km/s is expected, assuming a Hohmann transfer calculation);
- **Size** of the NEO targets must be larger than a given threshold (or brighter than a given absolute magnitude). A cut-off absolute magnitude of 19.5 is chosen, corresponding to objects larger than 300m in diameter (large enough for long-range acquisition by on-board optical sensors and to represent significant Earth impact damage potential, yet small enough to maintain a diverse sample of object types).

After filtering, a shortlist containing 15 objects was generated with a good cross-section of spectral and physical types retained. The shortlist was analysed in detail and 7 targets were selected by the mission study team in order to satisfy the primary mission objective of characterising the physical/chemical properties of different types of NEO in order of priority interest. These 7 target objects are summarised in Table 2, and are similar to those proposed by other authors^{6,7}. The bottom two asteroids (in grey) are considered as secondary reserve targets.

Table 2. Selected near-Earth asteroid targets for rendezvous by the SIMONE microsattellites

Priority	Target	Spectral type	Orbit class	Est. Diameter [km]	Comment
2 C types (common, poorly characterised, low density?, primitive)	1996 FG3	C	Apollo	1.3 effective	4-hour rotation Binary (16-hour orbit) Size ratio 0.31
	1989 UQ	CB	Aten	0.56 - 0.76	7.7-hour rotation
S-type (common, from inner main belt, higher temperature silicates)	1999 YB	S	Apollo	0.64	Only S-type left
M-type (metallic?, high density, X-types are P, M or E types)	2001 CC21	X	Apollo	0.39 - 1.1	Could be P, M, or E
	(4660) Nereus 1982 DB	X	Apollo	0.47 - 1.33	15-hour rotation C-type in some refs.
“Pot luck” (unknown type, select largest)	1994 CN2		Apollo	0.9 - 2.5	
Reserve (known parent body, Vesta)	(3361) Orpheus 1992 HR	V	Apollo	0.4	Parent body Vesta is Dawn mission target 4-hour rotation

The two selected X-type objects may be determined as either P, M or E types after further spectral follow-up observations are made. Thus, a good cross-section of different NEO spectral types can be characterised by the SIMONE rendezvous missions for risk assessment (C-types in particular are common but their basic properties are poorly understood) and science (detailed knowledge of primitive P or E-types, and metallic M-types). After C-type, M-type is a priority, but the nearest confirmed M-type metallic object is beyond the delta-V capability for a SIMONE rendezvous. Hence, a resonant orbit encounter strategy⁸ has been formulated by Telespazio for periodic brief flybys of the M-type asteroid Amun in the event that neither of the selected X-types are determined to be an M-type object. The C-type object 1996 FG3 is a high priority for science reasons, since it is a binary, i.e. two objects orbiting one another.

Science Objectives

The primary mission objectives of the SIMONE mission at each different NEO are to determine (in priority order):

- **Bulk density:** requiring both the mass and volume (size, shape) to be measured. For the particular spectral/physical class, it then allows predictions of the mass (and thus impact energy) to be made for other objects that are determined to be of the same class from ground observations. Bulk density can also be an indication of porosity;
- **Gravity field:** spherical/elliptical harmonics of the gravity field, together with a shape model, allow the derivation of large-scale internal density variations using a mass distribution model. These variations may have a bearing on the dynamical behaviour of a similar object on Earth approach, entry and impact, as well as

providing extra evidence as to the internal structure for aiding mitigation strategy development;

- **Surface topography/morphology:** the high-resolution surface information, in conjunction with compositional information, can be interpreted to give indications as to the object's internal structure. Surface features to examine include craters, grooves, fracture lines, regolith and boulders. From surface measurements, a detailed shape model will be constructed to improve mass and hence bulk density determination accuracy;
- **Composition:** provides spatial information to allow macroporosity to be estimated. Precise elemental/mineralogical composition can only be determined by a spacecraft encounter. Variations in composition across the surface will be correlated with topographic/morphological features, adding to the information available for assessment of the object's sub-surface properties/structure.

Science Payload

The baseline science payload instruments identified and selected to achieve these objectives are described in Table 3.

The total mass of the full baseline payload selection is only 13 kg (including margin). It has been found that all instruments can be accommodated on-board the SIMONE spacecraft design, all aligned along the same boresight (i.e. viewing from the same face) and within the available mass budget. Their power, data handling and communications requirements can also be fully satisfied. Further information on the science aspects of SIMONE is in preparation⁹.

Table 3. Baseline payload selection for each SIMONE microsatellite

Experiment	Measurement Objectives	Heritage	Performance
Multispectral Imaging System (MIS)	Size, shape, surface topography / morphology Contributes to bulk density, mineralogical composition, rotation state and binarity	SMART-1 mission (2003)	5.3°×5.3° field-of-view 1024×1024 pixel CCD 4-position filter wheel from visible to near-IR Resolution 1m at 11km
Radio Science Investigation (RSI)	Mass → bulk density Gravity field (J ₂)	2-way Doppler using X-band comms system Mars Express (2003), Rosetta missions	Range rate ~0.03mm s ⁻¹ over 100s Range 1-10m
X-Ray Spectrometer (XRS)	Elemental composition of most rock forming elements; potential for C detection	SMART-1 mission (2003)	2-10° field-of-view 0.5-10keV energies <150eV resolution
Near-Infrared Spectrometer (NIS)	Mineralogical composition across surface	SMART-1 mission (2003)	1.11mrad field-of-view 0.94-2.4µm wavelengths 6nm resolution
Laser Altimeter (ALT)	Size, shape, surface topography / morphology Contribution to mass, gravity field, bulk density	Clementine mission (1994) 1m resolution upgrade needed	<0.5mrad beam diverge 0.057° field-of-view

The Spacecraft

Spacecraft Design

The SIMONE microsatellites are 3-axis stabilised platforms with integrated avionics, an on-board Xenon ion propulsion system and an X-band communication system with an integral high gain antenna for TT&C data transfer at long ranges from Earth. Two lightweight high power solar array wings generate 1kW (maximum), sufficient to drive all subsystems and payload.

The autonomous attitude control system comprises 4 reaction wheels and 6 small hollow cathode thrusters for momentum dumping. On-board attitude determination is performed by redundant sun sensors, gyros and star trackers. The spacecraft has been designed to survive a high radiation exposure (dose damage and solar array degradation), the launch vibration environment and the extremes of the thermal environment expected during all 5 rendezvous missions over a maximum mission lifetime of 5 years. A summary specification is given in Table 4.

Table 4. Specification of the SIMONE microsatellite system design

Characteristic	Summary	Comments
Launch mass	120kg	Bus body dimensions of 600x600x710mm
Propulsion	T5 carbon-gridded ion engine	Specific impulse: 4,500s Nominal thrust: 18mN Lifetime: >30,000hours
Propellant	Up to 27kg of high purity xenon	Stored at 105 bar BoL
Power generation	1kW (BoL, AM0 and 1AU) 278W (EoL, AM0 and 2AU)	New hybrid array development using triple-junction GaAs cells
Power storage	150Whr (BoL)	Li-ion cells
Onboard computer	1750a (16bit machine, 2MIPS)	Radiation-hard, dual-redundant, autonomous FDIR
Data Storage	512Mbit 3D DRAM for payload data 4Mbit for housekeeping storage	Radiation-hard, dual redundant
Communications	X-band	0.5m HGA to 12 or 35m ground antenna
Uplink rate	Up to 300kbps	Selectable
Downlink rate	1.4kbps (minimum)	For a SIMONE-Earth distance of 2AU
Attitude control	Absolute pointing: 0.01° Relative pointing: 10arcsec in 10s Autonomous	3-axis using Momentum Wheels & Hollow Cathode Thrusters (HCTs). HCTs use the same propellant supply as the main engine
Attitude knowledge	Knowledge to better than 0.01°	Sun and Star Sensors

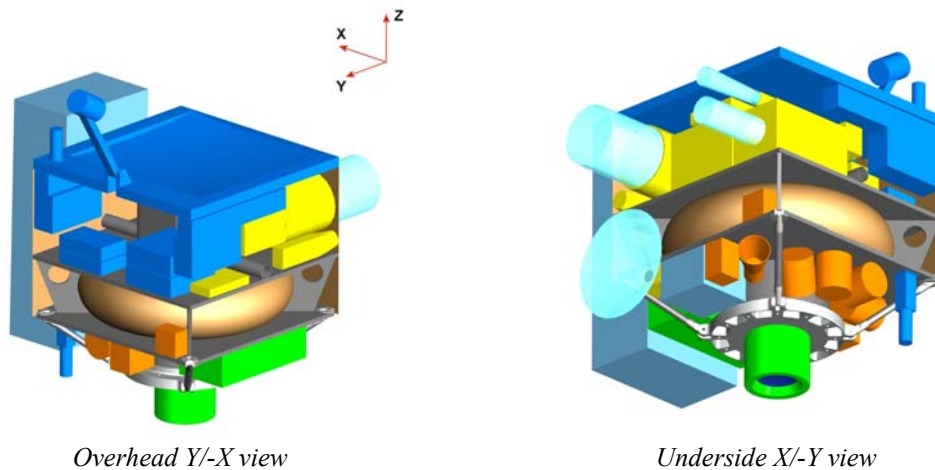


Figure 3. SIMONE spacecraft configuration (exterior panels and solar array wing removed for clarity)

A 3D CAD model of the SIMONE spacecraft is presented in Figure 3. This modelling was essential in order to perform a detailed accommodation analysis. It was established that all selected subsystem and payload equipment could fit within the tight volume constraints of the Ariane 5 ASAP launch. This was also necessary to complete a bus structure design that is not only lightweight, but also strong/stiff enough to secure equipment and survive the launch environment. To this end, finite element modelling of the structure (composite thrust tube, Al honeycomb/CFRP skin decks & panels, Al support struts and adaptor ring) was performed. All requirements were comfortably met within a total structure mass of 20kg.

Some innovative design was necessary to accommodate high volume equipment such as the pressurised Xenon fuel tank. Instead of using a conventional spherical tank, a toroidal tank fitted around the central thrust tube is used, which enables a minimum spacing between the upper & lower decks. The tank is an inner steel liner over-wrapped with composite material. Such tanks have already been developed by QinetiQ for terrestrial applications. To save further space, the High Gain Antenna forms the top panel of the bus and the feed is offset on a small deployable boom. The on-board computer and data handling system, and Xenon gas flow system are housed within the central thrust tube, along with the solar array drive mechanisms.

Thermal control is needed primarily for the Travelling Wave Tube Amplifiers (TWTAs) in the comms system located on the top deck with the payload, and the Xenon pressure tank. The ion thruster is thermally isolated from the bus and radiates heat to space. Heat needs to be dissipated when the TWTAs are operating. This is done by embedding heat pipes in the radiator panels on the solar wing sides of the bus, which are in contact with the TWTAs. Small heaters are used to keep the non-operating TWTAs and

pressure tank sufficiently warm at large solar distances. A thermal analysis was conducted to confirm the capability of the design within low mass.

All analyses indicate that the high performance microsatellite bus design proposed is feasible and can be built using current technology developments. Crucially, the ion propulsion and solar power systems can be integrated successfully to give the high power, high manoeuvrability capabilities.

Ion Propulsion System

The spacecraft design is dominated by the demanding total delta-V capability required to achieve a rendezvous with a typical NEO. The high specific impulse of the QinetiQ T5 gridded ion engine (4,500s), together with its relatively small size, makes it the ideal candidate for this mission, since a low propellant mass can be achieved and the engine can be accommodated within the spacecraft mass budget. The 10cm diam. T5 engine can be seen in Figure 4.



Figure 4. QinetiQ T5 ion thruster assembly

The xenon propellant required for a delta-V capability of 10 km/s is approximately 26kg on a 120kg spacecraft. With such a delta-V capability, nearly 60 NEOs become accessible for rendezvous by a SIMONE microsatellite. The flight times, fuel masses and engine on-times for each of the 5 SIMONE targets are given in Table 5. The T5 engine (with carbon acceleration grids) has a predicted total impulse capability of 3 million Ns and thrust range of 5-25mN. This equates to, for example, over 30,000 hours of operation at a constant 18mN of thrust, sufficient for the selected rendezvous targets (~18,000 hours). The thruster is baseline for the upcoming ESA GOCE mission, and is currently undergoing a comprehensive life testing/qualification programme.

Table 5. Ion propulsion performance per mission

Target	Flight time (yrs)	Xenon fuel mass (kg)	Engine time (hrs)
1989 UQ	2.67	25.0	17,030
2001 CC21	2.67	23.3	15,873
1996 FG3	3.55	22.2	15,124
Nereus	2.95	20.9	14,238
1999 YB	3.06	21.0	14,306

Lightweight High Power Solar Array

A solar array output of ~550W is required to power the T5 engine (for 18mN of thrust). Radiation degradation and the larger distances from the Sun during the encounter, increase the beginning of life power requirement to 1kW (measured at the Earth). Clearly, a lightweight deployable array, which when stowed is highly compact, is therefore also critical to the design. In order to significantly reduce the mass and volume of the solar array to within the launch constraints, a new array technology is being developed at QinetiQ whereby 28% efficient GaAs solar cells are attached to pre-tensioned membranes mounted within rigid tubular frames (see Figure 5).



Figure 5. QinetiQ lightweight high power solar array

Thick coverglass is used to reduce performance degradation due to radiation exposure, especially during the first month of the mission when the spacecraft is still passing through Earth's trapped proton belt at perigee. Each of the two steerable array wings have 11 panels and are 3.3m x 0.7m in area. When stowed, each wing occupies a volume of 0.3m x 0.7m x 0.2m. Deployment is controlled and synchronised. The total mass of the two wings is only 16kg. Alternative solar arrays such as the stretched lens arrays being developed under NASA funding are also being considered for use in the power system.

Mass Budget

The mass budget of each SIMONE spacecraft is based upon identification of known COTS components and associated suppliers able to meet the specification, QinetiQ technology development programmes, and the application of margin where existing items require modifications. The propellant mass includes a 10% margin and fuel for ADCS. Overall, the mass budget presented in Table 6 contains 11kg of margin and 6kg of redundant equipment. This enables dual redundancy of critical components, including: the on-board data handling unit (processor and memory); communications system (low gain antennas, transmitters, receivers and TWTAs); and the attitude determination & control system (gyros, star tracker heads and reaction wheels).

Subsystem	Mass (kg)	Mass Fraction (%)
ADCS	6.9	5.8
Propulsion	16.4	13.7
Power	23.3	19.4
Avionics	11.1	9.2
Mechanical	19.4	16.2
Payload	13.1	10.9
Dry mass total	90.2	75.2
Propellant *	29.8	24.8
Wet mass total	120	100

Table 6. SIMONE spacecraft mass budget

The Ground Segment

All SIMONE spacecraft will be identical in design and will have the same operational requirements for Telemetry, Tracking and Command (TT&C). Hence to lower recurring costs, a single mission control centre and a single ground station is foreseen to support the concurrent flight operations of all SIMONE spacecraft. Ground station costs are minimised by the use of a small 'near-Earth' ground station, transferring to leasing of link time on a commercial deep space network for deep space operations. In order to further reduce operations costs, the SIMONE spacecraft will have a degree of on-board autonomy to enable 'off-line' operations during non-critical periods of the missions and to minimise ground station usage. It is envisaged that a smaller

(and lower cost) near-Earth ground station will be used during the LEOP, check-out, Parking orbit and Earth escape phases of the missions. The spacecraft will be handed over to a larger deep space ground station to facilitate the required TT&C communications links with the spacecraft during the Interplanetary transfer, Rendezvous and Measurement phases of the missions. The ground segment architecture is illustrated in Figure 6.

The single mission control centre includes dedicated teams for the telemetry & command data handling of each SIMONE spacecraft. The mission control centre will be responsible for distribution of downlinked payload data from all SIMONE spacecraft to the science data centre for later processing and on-line distribution to the science community using established science data centres. Mission planning and flight dynamics functions provide simultaneous operational support to all SIMONE spacecraft. All SIMONE spacecraft will utilise the same system architecture and software, including that for orbit determination from tracking data, orbit and attitude manoeuvre planning, event sequencing, spacecraft system simulation, command generation for uplink, and telemetry data processing and monitoring. Re-use of existing systems, such as the ESA SCOS 2000 mission control software, is foreseen in order further reduce development costs.

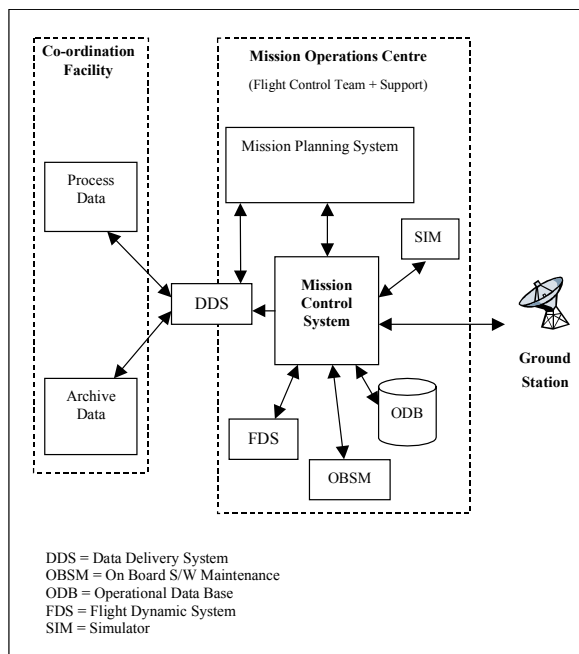


Figure 6. Ground segment architecture

The preferred baseline for spacecraft navigation is for a combination of radio and optical navigation. Radio navigation (using two-way Doppler tracking data at X-band) achieves a high precision range rate and is also used by the Radio Science Investigation during the Measurement phase at the target NEO. Optical

navigation (taking advantage of the Multispectral Imager payload instrument) is used for the NEO acquisition, approach and terminal guidance phase. This choice results from a compromise between the requirements of cost, complexity, onboard resource limitations and required accuracy. The image processing and orbit calculations associated with the navigation function are planned to be undertaken on the ground. Special software tools are required to perform the low-thrust interplanetary navigation, rendezvous and 'formation flying' manoeuvre calculations at the NEO target.

Conclusions

It is clear that to achieve an ambitious interplanetary mission, such as asteroid rendezvous, within 120kg mass and the confines of a microsatellite is a challenging goal. However, such a high-performance microsatellite bus is now feasible due to the emergence of a variety of key technologies including:

- Highly compact, yet capable, scientific instruments, enabling the baseline payload to be achieved within 13kg.
- A solar array (2 wings) that stows into a volume of 84 litres, has a mass of only 16kg but can generate 1kW using the latest triple-junction GaAs cells.
- A small gridded ion engine that is now a mature technology. In its latest variant (with graphite acceleration grids), the engine can exceed the necessary lifetime and total impulse requirements for an interplanetary mission like SIMONE.
- Small Hollow Cathode Thrusters that efficiently perform attitude control wheel momentum dumping direct from the main Xenon tank, rather than needing a separate cold gas thruster system.
- High-density, high-efficiency electronics and RF equipment (processors, memories, amplifiers, travelling wave tubes, etc.) that allow a flexible, capable data handling and communications architecture within a few kilograms.
- High-stiffness fibres and matrices that allow the construction of very lightweight spacecraft structures and pressurised gas propellant tanks with minimum mass.
- Small sensors and actuators (star cameras, sun sensors, momentum wheels and gyros) that offer high performance attitude and orbit control, but from devices that are <1kg each in mass.

Furthermore, the scale of economics with such a mission is a fraction of that normally associated with a larger, conventional mission. Significant cost savings arise from a rapid development schedule, a piggyback launch and commercial deep space ground station usage. However, the science return is not significantly compromised. Thus, it can be concluded that the "interplanetary microsatellite" is now not

only realisable in the short-term, but also a very cost-effective tool in solar system exploration. This can be exploited to make deep space more accessible with a single spacecraft, or to perform distributed exploration tasks with a network of multiple spacecraft as in the case of the SIMONE mission.

Outlook

With a significant delta-V capability of 10km/s provided by the T5 gridded ion propulsion system, near-Earth asteroids are not the only objects accessible to these high performance microsatellites. Mars Micro Missions are also being actively studied. Launched together onto a direct escape trajectory on a single low-cost Russian launcher, several such microsatellites can reach a low Mars orbit where they can form an orbiting constellation suitable for global atmospheric monitoring, communications data relay with surface assets and navigation services for other Mars missions. Slightly larger ion-propelled minisatellites are also being considered to carry larger payloads. Alternatively, the high performance, low-cost microsatellites may have high manoeuvre applications in Earth orbit, such as GTO-to-GEO transfer or plane changes in LEO for rendezvous and close inspection.

Acknowledgements

The authors would like to thank the other members the SIMONE mission study team for their contributions: Prof. Franco Bernelli, Mauro Massari of Politecnico di Milano; Dr. Ettore Perozzi of Telespazio; Martin Townend of SciSys; Martin Pätzold of University of Cologne; Glyn Dando, Mark Butcher, Matt Cosby, Steve Kynaston, Mike Price, Pete Jameson, Dave Mundy, Neil Wallace, Alan Freeman, Harj Jolly, Neil Redit, Chris Yates and Simon Clucas of QinetiQ.

This work was funded by the European Space Agency as part of preparatory activities for NEO space missions. The project was supervised by Andrés Galvez of the Advanced Concepts Team at ESTEC, Holland, with support from the Spaceguard Central Node at ESRIN, Italy.

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